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НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Кафедра конструкції літальних апаратів

ДОПУСТИТИ ДО ЗАХИСТУ

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«_____» _____ 2020 р.

ДИПЛОМНА РОБОТА
(ПОЯСНЮВАЛЬНА ЗАПИСКА)
ЗДОБУВАЧА ОСВІТНЬОГО СТУПЕНЯ
"БАКАЛАВР"

Тема: «Аванпроект ближньомісцевості літака вантажопідйомністю до 16 тонн»

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Київ 2020

MINISTRY OF EDUCATION AND SCIENCE OF UKRAINE

NATIONAL AVIATION UNIVERSITY

Department of Aircraft Design

AGREED

Head of the Department

Professor, Dr. of Sc.

_____S.R. Ignatovych

«____» _____ 2020 y.

DIPLOMA WORK

(EXPLANATORY NOTE)

OF ACADEMIC DEGREE

«BACHELOR»

Theme: «Preliminary design of short range aircraft with cargo capacity up to 16 tons»

Performed by: _____ M.R.Lutovinov

Supervisor: PhD, associate professor _____ T.P. Maslak

Standard controller: PhD, associate professor _____ S.V. Khizhnyak

Kyiv 2020

NATIONAL AVIATION UNIVERSITY

Aerospace Faculty

Department of Aircraft Design

Academic Degree «Bachelor»

Speciality: 134 "Aviation and Rocket-Space Engineering"

APPROVED

Head of the Department

Professor, Dr. of Sc.

_____S.R. Ignatovych

«____» _____ 2020 year

TASK

for bachelor diploma work

LUTOVINOV MAKSYM

1. Theme: «Preliminary design of short range aircraft with cargo capacity up to 16 tons» confirmed by Rector's order from 05.06.2020 year № 801/CT.
 2. Period of work execution: from 25.05.2020 year to 21.06.2020 year.
 3. Work initial data: cruise speed $V_{cr}=800$ km/h, flight range $L=1300$ km, operating altitude $H_{op}=11$ km, payload 16 tons.
 4. Explanation note argument (list of topics to be developed): choice and substantiations of the airplane scheme, choice of initial data; engine selection, aircraft layout, center of gravity position calculation, telpher design with carrying capacity up to 5 tons.
 5. List of the graphical materials: general view of the airplane (A1×1); layout of the airplane (A1×1); assembly drawing of the telpher (A1×1);
- Graphical materials are performed in AutoCad.

6. Calendar Plan

Task	Execution period	Signature
Task receiving, processing of statistical data	25.05.2020–26.05.2020	
Aircraft take-off mass determination	27.05.2020–29.05.2020	
Aircraft layout	30.05.2020–04.06.2020	
Aircraft centering determination	04.06.2020–08.06.2020	
Graphical design of the parts	08.06.2020–10.06.2020	
Completion of the explanation note	10.06.2020–13.06.2020	
Preliminary defence	13.06.2020–21.06.2020	

7. Task issuance date: 17.06.2020 y.

Supervisor of diploma work _____ T.P. Maslak

Task for execution is given for _____ M.R. Lutovinov

ABSTRACT

Explanatory note to the diploma work «Preliminary design of short range aircraft with cargo capacity up to 16 tons» contains:

sheets, 7 figures, 13 tables, 21 references and 3 drawings

Object of the design is a development of transport short range aircraft with cargo capacity up to 16 tons.

Subject of the design is the conceptual design of the upper cargo loading equipment such as telpher.

Aim of the diploma work is the preliminary design of short range cargo aircraft and conceptual design of telpher to provide loading of the cargo and move it in horizontal direction.

The method of design is analysis of the prototypes and selections of the most advanced decisions, the geometrical characteristics estimations, centre of gravity calculations, stress analysis of the drum axis.

Practical implementation of the results is defined by the results of the designing of the short range aircraft with carrying capacity up to 16. The materials of the diploma could be recommended for the airlines, which carry out cargo transportation and for the students of the aviation specialities.

AIRCRAFT, PRELIMININARY DESIGN, LAYOUT, CENTER OF GRAVITY POSITION, TELPHER.

List of diploma work

Format	№	Designation	Name	Quantity	Notes	
			<u>General documents</u>			
A4	1	NAU 20 09 L TP	Task of project	1		
			Graphic documentation			
		NAU 20 09 L 00 00 00 59	Cargo short range aircraft	2		
A1	2	NAU 20 09 L 00 00 00 59 GV	General view	1		
A1	3	NAU 20 09 L 00 00 00 59 AD	Aircraft layout	1		
A4	4	NAU 20 09 L 00 00 00 59 EN	Cargo short range aircraft			
			Explanatory note			
			<u>Special part documentation</u>			
	5	NAU 20 09 L 00 00 00 59	Upper Loading device			
A1	6	NAU 20 09 L 00 00 00 59 AD	Assembly drawing	1		
Department of Aircraft Design			NAU 20 09 L 00 00 00 59 EN			
Done by	Lutovinov M.R.		List of diploma work	list	sheet	sheets
Tutor	Maslak T.P.					
Instructor				402 AF 134		
N. contr.	Khizhnyak S.V.					
Head. of d.	Ignatovich S.R.					

CONTENT

Introduction.....	
1. Preliminary design of short range cargo aircraft with payload up to 16t.....	
1.1 The analysis of prototypes and choice of design parameters.....	
1.2. Aircraft geometry calculation.....	
1.2.1 Wing parameters.....	
1.2.2 Fuselage layout.....	
1.2.3 Tail unit parameters.....	
1.2.4 Engine selection for the designing aircraft.....	
1.2.5 Landing gear design.....	
1.3 Aircraft centre of gravity calculation	
1.3.1 Centre of gravity of equipped wing.....	
1.3.2 Centre of gravity of equipped fuselage.....	
1.3.3 Centre of centre of gravity for different types of capacity.....	
Conclusions to the main part.....	
2. Conceptual design of the upper loading device.....	
2.1 Description of the upper cargo loading device.....	
2.2 The type of rope design.....	
2.3 Hook assembly design.....	
2.4 The sheave and drum design.....	
2.5 Strength analysis of the drum axis	
2.6 Calculation of the braking device	
2.7 Electric engine parameters.....	
2.8 Gear box parameters.....	
2.9 Calculation of the attachment of the rope to the drum.....	
2.10 Movable parts of the lifting mechanism.....	
Conclusion to the special part.....	
General Conclusions.....	
References.....	
Appendices	

<i>Department of Aircraft Design</i>				<i>NAU 20 09L 00 00 00 59 EN</i>			
<i>Performed by</i>	<i>Lutovinov M.R.</i>			<i>Introduction</i>	<i>Letter</i>	<i>Sheet</i>	<i>Sheets</i>
<i>Principal</i>	<i>Maslak T.P.</i>						
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INTRODUCTION

Transport cargo aircraft is an aircraft designed to transportate goods of various types, sizes and purposes. The main feature of the cargo aircraft is the presence of huge cargo cabin, special cargo doors and hatches, rainforced floor and the precense of the cargo equipment which allows to load the cargo on board. Nowadays to deliver some types of cargo with the help of cargo aircraft is definitely the fastest way of transportation. If we are talking about the speed of delivery, the cargo aircraft in many times faster than trains and trucks. It is necessary to design such type of cargo aircraft which will have high level of reliability and regularity of flights, which will be in demand in the world market.

One of the main task of the aircraft design at the preliminary stage is to take into account all current general requiriments, such as flight safety; low fuel consumption; low noise level; low emissions, reliable operation. For freighters also are the list of specific requiriments: maximum cargo capacity, easy of cargo loading-unloading, efficiently cargo cabin space; the ability to take off and land in different runways; the usage of aircraft in different weather conditions; easy maintenance procedures, most favorable ration between cost and efficiency.

The aim of the diploma work is to perform the preliminary design of short-range cargo aircraft with maximum payload 16 tons. The designing aircraft is also based on these requiriments to provide conformity to the requirements of cargo aircraft. The main prototypes for the designing aircraft are An-148, An-178, An-12, CRJ-900, the main flight performances of the plane are cruise speed 800 km/h, flight range 1300 km, operating altitude 11 km.

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<i>Performed by</i>	<i>Lutovinov M.R.</i>			<i>Introduction</i>	<i>Letter</i>	<i>Sheet</i>	<i>Sheets</i>
<i>Principal</i>	<i>Maslak T.P.</i>						
<i>Adviser</i>					<i>402 AF 134</i>		
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<i>Head of dep.</i>	<i>Ignatovych S.R.</i>						

1. Preliminary design of short range cargo aircraft with payload up to 16 tons

1.1 The analysis of prototypes and choice of design parameters

The aim of the diploma work is to perform the preliminary design followed by the initial data. The first task of the work is data selection based on the analysis of prototypes, presented in the table 1.1. The prototypes for the future designed aircraft are aircraft in the short-range class: AN-148-100, AN-178, AN-12, and CRJ-900. According to the diploma work task, the designed aircraft must be short range transport cargo aircraft with cargo capacity up to 16 tons.

Statistical data of flight performances and geometrical parameters of the prototypes are shown in tables 1.1 and 1.2.

Table 1.1 - Operational and technical data of the prototypes

Parameter	Prototype			
	AN 148-100A	CRJ-900	AN-178	AN-12
Purpose of the aircraft	passanger	passanger	cargo	cargo
Crew/flight attend. persons	2/2	2/2	2/2	2/2
Maximum payload, kg	9000	10319	18000	13080
Cruise speed, km/h	820	848	830	570
Flight altitude, m	11000	12500	12200	10200
Maximum range, km	1920	2955	5500	3600
Thrust to weight ratio, N/kg	3,25	3,19	3,66	3,8
Takeoff distance, m	1580	1779	2500	1230
Landing distance, m	1600	1596	2300	1125
Take of weight, kg	38550	36514	51000	61000
Type of engine	2ТРДД	2ТРДД	2ТРДД	4ТВД
Take of thrust кN, кVТ	62,7	58,4	77,8	88,9
Pressure ratio	21	23,09	21	7,32
Bypass ratio	4,8	5,13	5,6	2,2

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Table 1.2 - Main geometrical parameters of prototypes

Parameter	Aircraft			
	AN 148-100A	CRJ-900	AN-178	AN-12
Fuselage shape	Circular	Circular	Circular	Circular
Fuselage diameter	3,35	2,69	3,55	4,5
Aspect ratio of fuselage	7,82	11,8	8,38	5,04
Sweep back angle 1/4 chord,	25	24	25	24
Wing aspect ratio	9,5	9	10	11,9

The next step of the design is the first weight estimation, optimization of the wingloading and choice of the engines. After the first iteration we could start to perform the preliminary design of wing, fuselage, tail and landing gear layout, we could find the necessary area of the wing to withstand payload, the area of the tail to provide stability of the aircraft, to calculate centre of gravity position of the aircraft.

On the base of the layout of prototypes the presented aircraft is high wing; with T-type tail unit; two turbojet engines located on the pylons under the wing, tricycle landing gear which retracts into the fuselage. Such scheme provides high efficiency of high lift devices; easy trim of the aircraft during flight; location of the tail unit, decreases destabilizing moment; reducing the areas of vertical and horizontal stabilisers. The initial data of the designing aircraft is shown in the Appendix A, received by the special computer program at the department of Aircraft Design.

1.2 Aircraft geometry calculation

1.2.1 Wing parameters

Considering the above choice of the parameters

The geometrical parameters of the wing are determined primarily based on the aircrafts take-off mass m_0 and the specific load which applied on the wing surface P_0 .

To find the area of the wing we use the formula:

$$S_w = \frac{m_0 \times g}{P_0},$$

where m_0 – take off weight, kg;

$g = 9,81 \text{ m/s}^2$ - acceleration of gravity;

P_0 – specific wing load during take off, Pa.

$$S_{wfull} = \frac{m_0 \cdot g}{P_0} = \frac{53122 \cdot 9.8}{4223} = 87.32[m^2]$$

Wing span can be calculated by the next formula:

$$l = \sqrt{S_w \cdot \lambda_w} = \sqrt{87.32 \cdot 9.58} = 28.92[m]$$

where λ_w – aspect ratio of the wing.

The root chord and tip chord of the wing can be obtained using the next formulas:

$$b_o = \frac{2S_w \cdot \eta_w}{(1 + \eta_w) \cdot l} = \frac{2 \cdot 87.32 \cdot 4.05}{(1 + 4.05) \cdot 28.92} = 4.84[m]$$

where η – taper ratio;

$$b_t = \frac{b_o}{\eta_w} = \frac{4.84}{4.05} = 1.19[m]$$

The position of the front spars and rear spar to the leading edge of the wing is taken according to the relative coordinations are $\bar{X}_1 = 0,2$; $\bar{X}_2 = 0,6$.

The relative coordinates for the root chord are:

$$X_1 = \bar{X}_1 \times b_0 = 0,2 \times 4,84 = 0,968[m]$$

$$X_2 = \bar{X}_2 \times b_0 = 0,6 \times 4,777 = 2,904[m]$$

The relative coordinates for the tip chord are:

$$X_1 = \bar{X}_1 \times b_k = 0,2 \times 1,19 = 0,238[m]$$

$$X_2 = \bar{X}_2 \times b_k = 0,6 \times 1,18 = 0,714[m]$$

Determination of the mean aerodynamic chord of the wing (MAC, b_{mac}) is conducted with the help of geometrical method (figure 1.1). So, $b_{mac} = 3,341$ m.

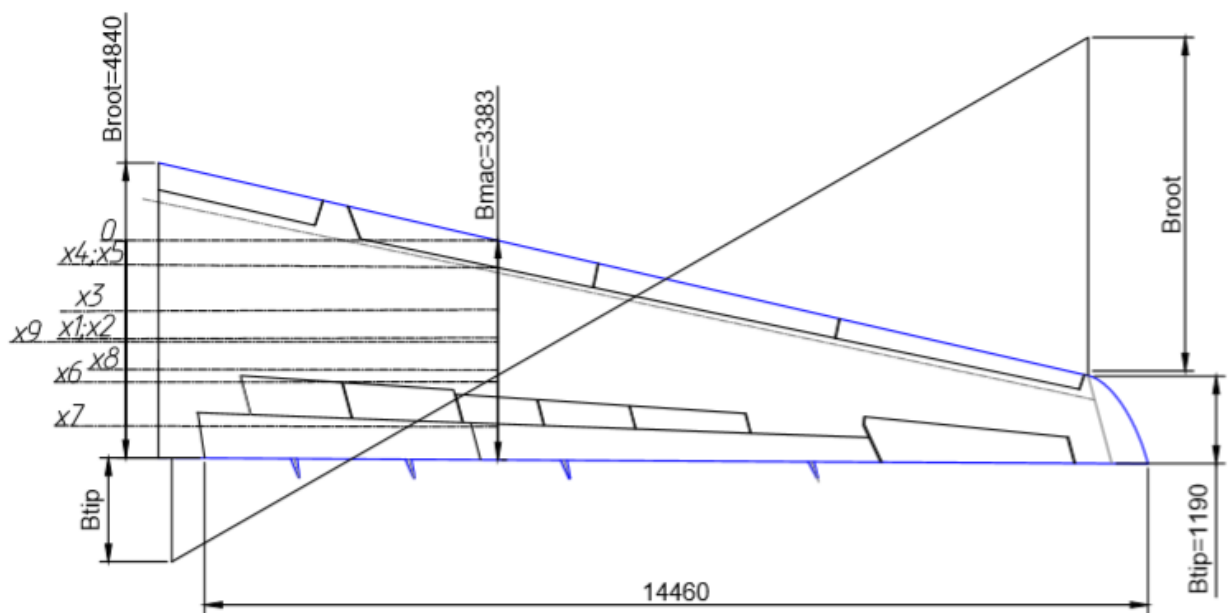


Figure 1.1 - Mean aerodynamic chord of the wing

Based on the analysis of statistical data of prototype aircrafts, as well as based on the accepted initial data, we have chosen the relative thickness of the airfoil equal to $\bar{c}_i = 0.118$

In result the airfoil thickness in the root part and in the tip part are equal:

$$c_{root} = \bar{c}_{root} \times b_0 = 0,118 \times 4,84 = 0,571[m]$$

$$c_{tip} = \bar{c}_{tip} \times b_0 = 0,118 \times 1,19 = 0,14[m]$$

For the designing aircraft the slats and double-slotted flaps are taken as a high lift devices without aerodynamic balance. The main task of the high lift devices is to provide necessary lift force during take off and landing.

The slats will take full length of the leading edge of the wing. The area of the slats could be found by the equation: $S_{slat} = 0.1 S_{wing}$

$$S_{slat} = 0.1 \times 87,42 = 8,742[m]$$

The area of the flaps could be found by the equation $S_{flaps} = 0.17 S_{wing}$

$$S_{flaps} = 0.17 \times 87,42 = 14,86[m]$$

The trailing edge of the wing is filled not only by flaps and also by the ailerons at the tip.

To find aileron span we use the equation:

$$l_{ail} = 0.375 \cdot \frac{l_w}{2} = 0.375 \cdot \frac{28.92}{2} = 5.72[m]$$

The aileron area:

$$S_{ail} = 0.065 \cdot \frac{S_w}{2} = 0.065 \cdot \frac{87.32}{2} = 2.83[m^2]$$

The range of aileron deflection is upward $\delta_{ail} \geq 20^\circ$; downwards $\delta_{ail} \geq 10^\circ$

1.2.2 Fuselage layout

It is necessary to follow the requirements of the aerodynamics to choose the correct shape and size of the designed fuselage. There are requirements of aerodynamics such as streamlining and cross-sectional area.

If we are talking about subsonic transport aircrafts ($V < 800$ km/h) shock wave drag is not important. Therefore, the shape should be chosen from the condition to provide the lowest values according to the friction resistance C_{xf} and form drag C_{xp} . On the transonic and supersonic flights the shape of the nose part of the fuselage affects on the value of form drag C_{xb} .

For transonic aircrafts the nose part should be calculated as:

$$L_{ns} = (2 \dots 3) D_f, \text{ where } D_f - \text{fuselage diameter.}$$

Also, it is necessary to take into account strength requirements.

To provide minimum weight the best variant of fuselage cross section is round cross section. Due to this, the thickness of the fuselage skin will be the lowest.

The geometrical parameters are: fuselage diameter D_f ; fuselage length L_f ; - fuselage fineness ratio:

$$\lambda_f = \frac{D_f}{L_f}.$$

- nose part fineness ratio:

$$\lambda_{ns} = \frac{L_{ns}}{D_f}.$$

- tail part fineness ratio:

$$\lambda_{tp} = \frac{l_{tp}}{D_f}.$$

where l_{np} and l_{tp} – the lengths of nose part and tail part. The length of the fuselage is determined taking into account the scheme of the designed aircraft.

The following fuselage parameters are:

$$l_f = \lambda_f \cdot D_f = 8 \cdot 3.35 = 26.8[m]$$

$$l_{np} = \lambda_{np} \times D_f = 3,1 \times 3,35 = 10,3[m]$$

$$l_p = \lambda_p \times D_f = 2,1 \times 3,35 = 7,035[m]$$

Also it is necessary to make the minimum mid-section of the fuselage S_{ms} . For cargo aircraft the mid-section depends from size of cargo cabin.

Crew cabin should take as less as possible area on the aircraft. However, on the other hand it must provide comfortable conditions for crew operation and rest. The crew's workplaces must meet strict requirements. The pilot's places must provide good overview from the cabin. The dimensions of the crew cabin depend on the number of pilots. Nowadays the flight crew consists from 2 or 3 pilots.

The crew contains of: the captain (first pilot), co-pilot. The pilots are seated in seats next to each other.

The flight crew compartment is separated from other compartments by a partition with a lockable door.

Flight attendant can be located in the crew cabin or outside of it. Flight attendants must have separate seats with safety harnesses. The number of the flight attendants depends from the class of aircraft and type of cargo.

The crew cabin is designed similar to the cabin of prototype aircraft.

1.2.3 Tail unit parameters

The main task during calculation of aerodynamic layout is to choose the location of the empennage. To provide the longitudinal stability of the aircraft it is necessary to locate the center of gravity at the front of the aircraft focus.

The distance between the aerodynamic centre of the tail surfaces and to the centre of gravity of the aircraft, located at the mean aerodynamic chord (MAC) of tshows the rate of longitudinal stability.

Moment of longitudinal stability could be found:

$$m_x^{Cy} = \bar{x}_T - \bar{x}_F < 0$$

where m_x^{Cy} moment coefficient, x_T, x_F , - center of gravity and aircraft focus. The aircraft has neutral longitudinal static stability if the m_x^{Cy} equals 0.

Based on the analysis of statistical data of prototype aircrafts, we have:

$$S_{HTU} = \frac{b_{MAC} \cdot S_w}{L_{HTU}} \cdot A_{HTU} = \frac{3.38 \cdot 87.32}{9.46} \cdot 0.55 = 18.87[m^2]$$

$$S_{VTU} = \frac{l_w \cdot S_w}{L_{VTU}} \cdot A_{VTU} = \frac{28.92 \cdot 87.32}{9.46} \cdot 0.09 = 21.15[m^2]$$

The length of nose part and tail part, sweep back angle, location of the wing, stability and controls of the aircraft effects on the length of vertical and horizontal stabilizer.

In the first approximation, we can assume that, $L_{HTU} \approx L_{VTU}$.

Determination of elevator and rudder areas. The area of elevator is usually calculated as:

$$S_{el} = 0.2765 \cdot S_{HTU} = 5.21[m^2]$$

The area of rudder is usually calculated as:

$$S_{rud} = 0.2337 \cdot S_{VTU} = 4.94[m^2]$$

The wing span and empennage are connected by static dependence.

$$L_{HTU} = (0.32 \dots 0.5) l_w = 0.323 \times 28.95 = 9.3[m]$$

The height of vertical stabilizer determines due to relation of the of wing location due to fuselage and location of the engines under the wing. So, we have:

Taper ratio for horizontal and vertical stabilizers should be chosen as:

$$\eta_{HTU} = 2...3 \text{ and } \eta_{VTU} = 1...3,3.$$

Based on the analysis of statistical data of prototype aircrafts we take:

$$\eta_{HTU} = 2,51 \text{ and } \eta_{VTU} = 1,367$$

For subsonic aircrafts the aspect ratio should be chosen as:

$$\lambda_{VTU} = 0,8...1,5 \text{ and } \lambda_{HTU} = 3,5...4,5$$

Based on the analysis of statistical data of prototype aircrafts we have:

$$\lambda_{VTU} = 0,95 \text{ and } \lambda_{HTU} = 4,5$$

The calculation of the b_{tip} , b_{root} calculated due to following formulas:

For horizontal tail unit it is calculated as:

$$b_{0HTU} = \frac{b_{HTU}}{\eta_{HTU}} = \frac{2.89}{2.51} = 1.15[m]$$

$$b_{0HTU} = \frac{2S_{HTU} \cdot \eta_{HTU}}{(1 + \eta_{HTU}) \cdot l_{HTU}} = \frac{2 \cdot 18.87 \cdot 2.51}{(1 + 2.51) \cdot 9.32} = 2.89[m]$$

For the vertical tail unit we can use such formula:

$$b_{0VTU} = \frac{b_{VTU}}{\eta_{VTU}} = \frac{5.32}{1.367} = 3.89[m]$$

$$b_{0VTU} = \frac{2S_{VTU} \cdot \eta_{VTU}}{(1 + \eta_{VTU}) \cdot l_{VTU}} = \frac{2 \cdot 21.15 \cdot 1.367}{(1 + 1.367) \cdot 4.59} = 5.32[m]$$

Relative thickness of airfoil in the first approach will be $\bar{C}_{em} \approx 0,8 \bar{C}_w$.

If we attach the horizontal stabilizer on the fin, we assume $\bar{C}_{em} = (0,08...0,1)$.

For the designing aircraft we take $\bar{C}_{em} = 0,1$.

The sweep back angle of empennage is taken near $3...5^\circ$ more than the

sweep angle of the wing. This performed to ensure the controllability of the aircraft when the form drag will appear on the wing. $\chi_{HTU}=32^\circ$; $\chi_{VTU}=40^\circ$.

1.2.4 Engine selection for the designing aircraft

Observing the thrust-to-weight ratio of the prototypes, a value of 3,2 is chosen for the designing aircraft. On the base of the necessary thrust required and parameters of the engine selected earlier in the initial data, we select the bypass turbojet engine D436T1 (figure 1.2), characteristics of this engine are shown in table 1.4.

Table 1.4 - D436T1 engine characteristics

№ п/п	Engine data	Units of measurement	Value
1	Type of engine	-	D436T1
2	Take off power	kN	73,57
3	Power in cruise flight mode	kN	14,71
4	Specific fuel consumption on take off mode	kg/N×hour	0,037
5	Specific fuel consumption in cruise flight mode	kg/N×hour	0,062
6	Engine pressure ratio	-	24
7	Dry engine mass	kg	1450
8	Bypass ratio	-	4,95



Figure 1.2 – Turbojet engine D436T1

1.2.5 Landing gear design

A retractable tricycle landing gear is chosen. It is the most commonly used landing gear. During the takeoff all weight of the aircraft is taken by the main wheels, and also such scheme has better lateral stability on the ground.

During designing of aircraft, when we know the the center of gravity position and we have no general view of designed aircraft, we can calculate next parameters of landing gear:

The distance from the centre of gravity to the main landing gear approximately could be found:

$$e = 0.2673 \cdot b_{MAC} = 0.2673 \cdot 3.38 = 0.903[m]$$

Landing gear wheel base is calculated as follows:

$$B = 0.4526 \cdot L_f = 0.4526 \cdot 26.8 = 12.129[m]$$

The nose landing gear can carry 6-10% of full aircraft weight, so nose landing gear can be calculated as follows:

$$d_{ng} = B - e = 11.226[m]$$

Wheel track of designing aircraft calculated as follows:

$$T = 0.6072 \cdot B = 0.6072 \cdot 12.129 = 7.36[m]$$

The size of landing gear wheels depends on the loads during take-off and landing. For the nose landing gear support was considered dynamic loads.

Pneumatics type and the pressure, depends on the runway where the designed aircraft will be used.

The loads which will applied to the wheels can be determined as follows:

For the nose landing gear the wheel load is equal:

$$P_{NLG} = \frac{(9.81 \cdot e \cdot k_g \cdot m_0)}{(B \cdot z)} = \frac{(9.81 \cdot 0.903 \cdot 1.8 \cdot 53122)}{(12.129)} = 71924.008[N]$$

For the main landing gear wheels load is equal:

$$P_{MLG} = \frac{(9.81 \cdot (B - e) \cdot m_0)}{(B \cdot n \cdot z)} = \frac{(9.81 \cdot (12.129 - 0.903) \cdot 53122)}{(12.129 \cdot 2 \cdot 2)} = 219240.5[N]$$

Where K_g is dynamics coefficient and equals 1.5-2.0.

The dimensions of tires for designing aircraft is shown in the table 1.5.

Table 1.5 - Tires characteristics

Main gear		Nose gear	
Tire size	Ply rating	Tire size	Ply rating
1244x 431	32	990x330 mm	16

The airplane tires are inflated to high pressure, near 13 bar (200 psi, 13 atm).

1.3 Aircraft centre of gravity calculation

At the preliminary design stage, after the first calculation of the geometrical parameters, after the choice of the schemes of the aircraft parts, the next task is to estimate the weight of the main components of the aircraft, based on statistic data. The weight of the aircraft is divided into the empty weight of the aircraft and the weight of fully loaded aircraft. It is a normal practice to combine the equipment of the aircraft, various components and systems.

The general requirements for the layout of the aircraft are following: each unit (cargo) of the aircraft must be located in such way, where in will be the most useful; the layout of the aircraft should provide suitability of control and maintenance of systems and units, and also to provide repairing and installation of parts and units; the layout should provide convenience of the general assembling of

structure of aircraft; structural layout should provide less weight with sufficient strength and durability.

1.3.1 Centre of gravity of equipped wing

Mass of full wing structure, mass of the equipment which is mounted in the wing, fuel, are included in the mass of equipped wing. The beginning of the the center of gravity coordinates is choosed in the projection point of MAC to the plane. Positive values of these coordinates moves to the tail part of aircraft. Centering information about equipped wing is shown in table 1.6.

Table 1.6 - Centering of equipped wing

N	Name	Mass		C.G. coordinates x_i (m)	Moment $m_i x_i$ (kgm)
		Units	total mass m_i (kg)		
1	Wing (structure)	0.11993	6370.92	1.4196	9044.16
2	Fuel system, 40%	0.00152	80.75	1.4196	114.62
3	Control system, 30%	0.00219	116.33	1.014	117.96
4	Electrical equip. 10%	0.00208	110.49	0.338	37.34
5	Anti-icing system 70%	0.01197	635.87	0.338	214.92
6	Hydraulic system, 70%	0.01358	721.39	2.028	1462.99
7	Power units	0.08915	4735.82	2.65	12549.93
8	Equipped wing without fuel and LG	0.24042	12771.59	1.8433	23541.95
9	Fuel	0.13088	6952.60	1.4534	10104.91
	Equipped wing	0.3713	19724.19	1.7058	33646.87

Coordinates of the center of gravity of equipped wing is calculated as the ratio of sum of mass moments to sum of masses:

$$X'_w = \frac{\sum m'_i x'_i}{\sum m'_i}$$

1.3.2 Centre of gravity of equipped fuselage

The beginning of the coordinates choosed starts in the projection from the front part of the fuselage on horizontal axis. Axis X is horizontal line of fuselage. Center of gravity coordinates can be calculated as follows:

$$X_f = \frac{\sum m_i' X_i'}{\sum m_i'};$$

When we determined the center of gravity of equipped wing and the fuselage, we can compose following equation:

$$m_f x_f + m_w (x_{MAC} + x_w') = m_0 (x_{MAC} + C) .$$

The loacation of the front part of MAC due to the nose part of fuselage , the value X_{MAC} can be calculated as follows:

$$X_{MAC} = \frac{m_f x_f + m_w \cdot x_w' - m_0 C}{m_0 - m_w} = 10.9 .$$

where: m_f - mass of equipped fuselage.

m_w - mass of equipped wing.

C – the lengrh from the front of MAC to the center of masses of aircraft.

$$C = (0,28...0,32) \cdot b_{mac}$$

The values of center of masses of equipped fuselage shown in table 1.7.

Table 1.7 - Centre of gravity of equipped fuselage

	Objects	Mass		Coordinates of C.G., m	Mass moment (kgm)
		Units	Total (kg)		
1	Fuselage	0.12476	6627.5	13.41	88881.41
2	Horizontal tail unit	0.0174	928.04	0.91	844.51
3	Vertical tail unit	0.02031	1078.9	2.07	2233.33
Equipment					
4	Anti-icing system, 15%	0.00256	135.99	21.45	2917.03
5	Air-conditioning 15%	0.00256	135.99	13.41	1823.79
6	Heat and sound isolation	0.0062	329.35	13.41	1823.79
7	Control system, 70%	0.00511	271.45	13.41	3640.46
8	Hydraulic sys30%	0.0058	308.1	18.77	5783.17
9	Electrical eq, 90%	0.0187	993.38	13.41	13322.23
10	Radar	0.0059	313.41	1.081	338.8
11	Air-navig. system	0.0088	467.4736	1.88	878.85
12	Radio equipment	0.0044	233.73	1.081	252.66
13	Instrument panel	0.0103	547.15	2.58	1411.66
Cargo Aircraft					
14	Seats of crew	0.00056	29.96	2.78	83.29
15	Seats for accomp.person	0.00037	19.97	3.6	71.9
16	Cabin equipment	0.0006	31.87	6.26	199.52
17	Cargo equipment	0.02845	1511.32	23.59	35655.08
18	Non typical eq.	0.0037	196.55	896	1761.1
19	Service equipment	0.01594	846.76	6.99	5918.88
20	Nose landing gear	0.00297	157.77	2.78	438.6
21	Main landing gear	0.04153	2206.15	12.4	27356.34
	Equipped fuselage without payload	0.26286	13963.94	11.65	162754.77
Payload					
22	Cargo	0.295	15670.99	13.6	213125.46
23	Crew	0.00564	299.6	2.78	832.91
	Total	0.999	53122	18.25	608589.72

1.3.3 Calculation of centre of gravity for different types of capacity

At this stage of designing of aircraft we already have the location of the wing with relation to the aircraft fuselage on the designed drawing. We already have calculated the centers of gravity for the fuselage and wing. So we need to calculate the centers of gravity for different types of aircraft loading and regimes of flight.

Center of gravity of aircraft it is the relative location of CG position related to the MAC, and this relation showed in percentage. Calculations of the center of gravity masses is shown in tables 1.8 and 1.9.

Table 1.8 - Centre of gravity summary list

Name	Mass m_i , kg	Coordinates, m	Mass moment, kgm
Equiped wing without fuel and landing gear	12771.59	13.84	176867.3
Nose landing gear (retracted)	157.77	1.78	280.83
Main landing gear (retracted)	2206.15	12.4	27356.34
Fuel	6952.6	12.37	86003.75
Equiped fuselage	13963.64	11.65	162754.77
Cargo	15670.99	13.6	213125.46
Crew	299.6	2.78	832.91
Nose landing gear (opened)	157.77	2.78	438.6
Main landing gear (opened)	2206.15	12.4	27356.34

Table 1.9 Centre of gravity for different aircraft modes

No	Variants of the loading	Mass, kg	Moment of the mass, kg m	Centre of the mass, m	Centering, %
1	Take-off mass (landing gear opened)	53122	664159.15	12.76	22.53
2	Take-off mass (landing gear retracted)	53122	664121.38	12.76	22.51
3	Landing variant (landing gear opened)	48422.2	603382.24	12.46	15.48
4	Transportation variant (without payload)	36351.38	454095.91	12.49	16.39
5	Parking variant (without fuel and payload)	29099.16	367417.02	12.62	18.37

Conclusions to the main part

The preliminary design of the short range cargo aircraft with cargo capacity up to 16 tons was developed in this part. The main parts and elements of the designed aircraft were considered.

So, in the results of this part of diploma work we have the preliminary design of aircraft with wing span 28.92 m, the designed length of the fuselage is 26.8 m with fuselage diameter 3.35 m. To provide thrust for all regimes of flight, we choose the D436T1 engine, because of low fuel consumption, low weight to thrust ratio and low levels of emissions and noise. Also the center of gravity positions were calculated. The most forward centre of gravity position is 15,48% from the leading edge of the mean aerodynamic chord, and the most aft centre of gravity is 22,53% for the takeoff regime. Due to these calculations we can confirm that all masses of the aircraft are in equilibrium.

2. Conceptual design of the upper loading device

2.1. Description of the upper cargo loading device

Top cargo loading equipment is designed for loading and unloading equipment without wheels, containers or pallets and mono-loads.

This load device is designed to load the cargo from the cargo area and move the cargo along or across the cargo cabin of the aircraft. Lengthwise displacement of loading device is carried out on rails mounted on the ceiling of the cargo compartment. The transverse movement of cargo is carried out by two trolleys mounted on the frame of on board loading loader. Each trolley is equipped with a telpher to load the cargo up to 5 tons and the mechanism of movement. To prevent the breakage of the telpher, there is weight limiter, which do not permit to load the cargo more then 5 tons.

A monorail is installed on the fuselage ceiling in the aircraft along which a telpher moves, designed to perform loading and unloading operations.

The mechanisms of the overhead crane provide three movements: lifting the load, moving the trolley with a telpher and moving the full crane. The lifting mechanism is a telpher associated with a double tackle.

When choosing this device, you need to fully understand the conditions under which it will be used.

General characteristics of telfers:

- Loading capacity fluctuates from 0,6 to 100 tons;
- Maximum lifting height up to 100 meters;
- Speed of lifting of freight from 4 to 16 m / min.

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<i>Performed by</i>	<i>Lutovinov M.R</i>			<i>Special part</i>	<i>Letter</i>	<i>Sheet</i>
<i>Supervisor</i>	<i>Maslak T.P</i>					
					<i>402 AF 134</i>	
<i>Stand.contr.</i>	<i>Khizhnyak S.V</i>					
<i>Head of dep.</i>	<i>Ignatovych S.R.</i>					

There are different types of telfers: a) movable, c) stationary, b) explosion proofed, d) with decreased overhall height (Figure 2.1).

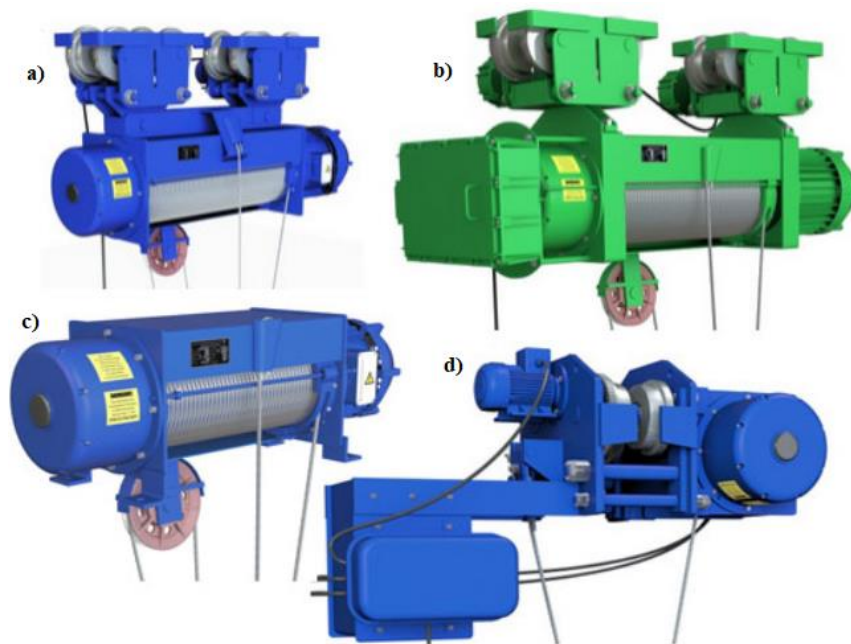


Figure 2.1 – Types of telfers.

Their carrying capacity can be up to 50 tons. The telfers can be with normal height, and can be reduced, they can be designed for different climate conditions, they are also different by the type of loading.

Telfers are an independent device, they can lift loads vertically, and then move them in the horizontal direction.

Telfers allow to load the specific cargo quickly without additional help. They are prepared for the work in conditions of "crisis" temperature - in the range from -40°C to $+40^{\circ}\text{C}$.

There are two main types of telfer design, in particular, the rope and chain devices are used in telfers. The telfers can be stationary or mobile, and so on.

Based on the analysis of statistical data, the designed telfer should follow next characteristics: carrying capacity - 5 tons; lifting speed – 20 m/min; lifting height - 3.5 meter, moving speed – 25 m/min

The general view of the designed telfer is presented on the drawings and in the figure 2.2.

The main parts of it are drum (1), electric motor for movement (2), speed reducer for movement (3), speed reducer for lifting (4), electro motor for movement (5), wheels (6), rope (7), hook (8), rope balancer (9).

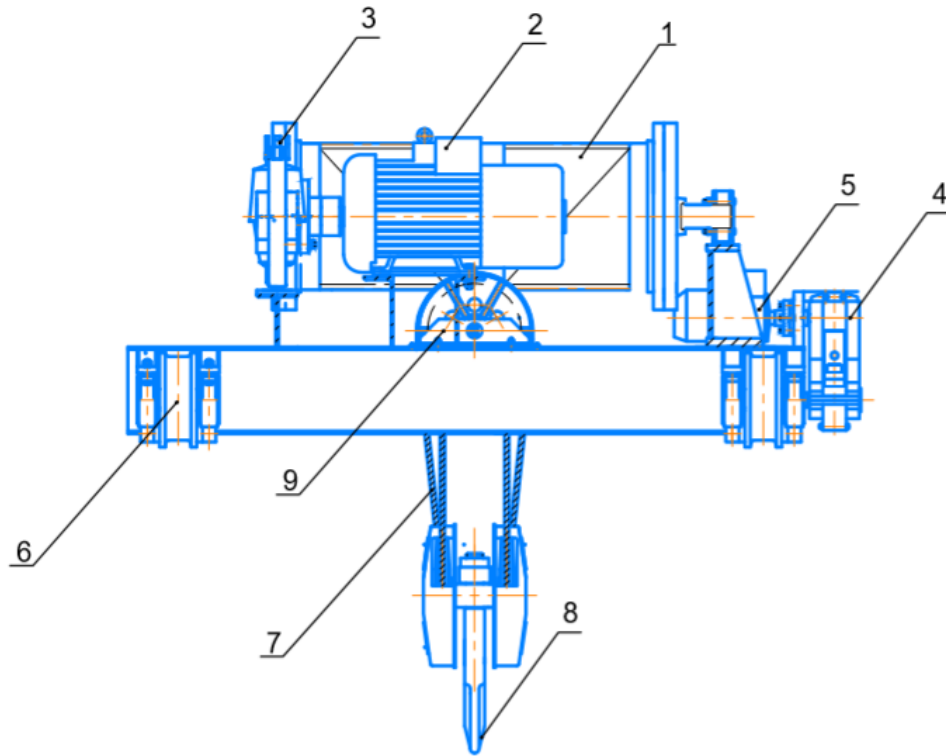


Figure 2.2. Drawing of the lifting mechanism

2.2. The type of rope design

The rope of the telpher is one of the most important elements of the cargo loading device which affects on the safety of the loading and unloading of the cargo. The ropes for the telpher are similar as lifting equipment.

A correctly selected and mounted rope will ensure minimal wear of both the rope itself and the drum, to which the rope is attached.

It is necessary to calculate the maximum tension of the choosing rope with the help of the following formula:

$$F_{\max} = \frac{G}{Z_{n.r.} u_n \eta_n \eta_{n.b.}} \cdot$$

where, G – weight of cargo =50 кN;

$Z_{n.r.}$ – number of ropes which spooled on the drum, $Z_{n.r.}=2$;

u_n – mechanical advantage of block and tackle,

η_n – efficiency of block and tuckles,

$\eta_{n.b.}$ – efficiency of rope balancer.

We choose $Z_{n.r.}=2$; $u_n=4$; $\eta_n=0,94$; $\eta_{n.b.}=0,96$.

$$F_{\max} = \frac{50}{2 \cdot 4 \cdot 0,94 \cdot 0,96} = 6,9[kN]$$

The type of the rope we will choose from the next conditions:

$$F_{\max} z_p \leq F_{break} .$$

Z_p – is coefficient for the movable ropes for M2 lclassification of mechanism, and it is shows in the table 2.1; where F_{break} – breakage force of the rope.

Table 2.1 - Rope safety factors

Class of mechanism	Movable ropes	Stationary ropes
	Z_p	
M1	3.15	2.50
M2	3.35	2.50
M3	3.55	3.00
M4	4.00	3.50
M5	4.50	4.00
M6	5.60	4.50

According to the [15] there are six groups of operating modes, each of them is characterized by a certain combination of classes for mechanism usage and

applied loads on the mechanisms.

The correct choice of the rope classes provides the normal operating time of the mechanism. Approximate characteristics of the mechanism classes are shown in the table 2.2, the characteristic of their loading classes are in the table 2.3.

Table 2.2 – Loading mechanism classes

Class of usage	Use relative to time
A0	Rarely used
A1	Irregular
A2	Regular with low intensity
A3	Regular with mid intensity
A4	Irregular or regular (double shifted work)
A5	Intensive with (3 shifted work time)
A6	Very intensive (3 shifted work time)

Table 2.3 - Classes of loading

Class of loading	Characteristic of loading	
	Lifting mechanism	Horizontal movement
B1	Minimal loads prevail; maximum loads are rare.	Rare starts and brakes, significant work and idle runs; low mass of cargo, low intensity production
B2	Medium and minimal loads prevail; maximum loads are relatively rare.	Rare but regular starts and brakes, small work and idle runs; insignificant mass of cargo.
B3	Medium and minimal loads prevail; minimum loads are episodic in nature.	Regular starts and braking, small work and idle runs, relative large mass of cargo.
B4	Maximum loads prevail and are close to maximum; average loads are episodic	Frequent starts and braking with rare pauses to the rest, large mass of cargo

Due to these tables we choose next characteristics: A1 – irregular working times, it is usual situation for the cargo aircraft and B3 – working with medium loads.

Now we can choose the classification of mechanism. Table 2.4 shows classification of mechanisms and the characteristics for loading devices.

Table 2.4 - Classification of load mechanisms

Class of usage	Servicelife, h	Class of loading			
		B1	B2	B3	B4
A0	Up to 800	1M	1M	1M	1M
A1	From 800 to 1600	1M	1M	2M	3M
A4	From 6300 to 12500	3M	4M	5M	6M

Due to this table we can assume that our loading device is 2M of classification mechanism. With approximately 1600 hours lifetime. For the classification mechanism 2M- Z_p will be equal 3.35. This classification is needed to analyze servicelife of the designed mechanism, under special conditions of work.

$$6,9 \cdot 3,35 = 23,12 \text{ kN} \leq F_{break} .$$

From the following conditions from [16] we choose the two lay rope type ЛК-Р (Figure 2.3) construction with following characteristics:

6,2-Г-1ОЖН1960, the diameter equals $d_r=6,2$ mm. Letter Г means that this rope can be used in cargo mechanisms, Н means that it is non-rotating, ОЖ – zink-coated wire, 1960 marking group.

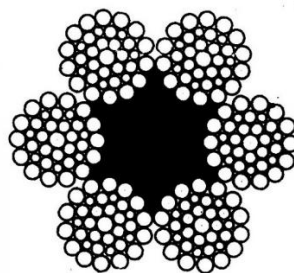


Figure 2.3 - Cross-section of the choosed rope

This rope has a large number of small-diameter wires and high fatigue resistance when it bends on blocks.

2.3. Hook assembly design

Due to the lifting capacity 5 tons of our cargo lifting device, and the classification of mechanism M2, due to the [17] we choose No 14 of hook.

It is necessary to calculate the height of the nut which mounted to the ends of the hook, due to the [17] the nominal diameter of the thread for such type of hook is M48 with the help of the following formula:

$$H = \frac{G}{\pi d_1 k_1 k [\tau]}.$$

where d_1 – inner diameter of the thread;

k_1 – ratio of the height of the section of the thread to the pitch S , for the metric thread;

k – the coefficient of load distribution between the pitch of thread,

$d_1=41,9$ mm, $k_1=0,87$, $k_{un}=0,56$;

$[\tau]$ – allowable shear stress,

$[\tau]=0,15 \cdot \sigma_T=0,15 \cdot 240=36$ MPa ($\sigma_T=240$ MPa – yield strength for steel 20).

So, the results are:

$$H = \frac{50 \cdot 10^3}{3,14 \cdot 41,9 \cdot 0,87 \cdot 0,56 \cdot 36} = 22 \text{ [mm]}$$

To design of the hook block, it is necessary to calculate the block diameter:

$$D_{bl} \geq h_2 d_r.$$

where h_2 – coefficient of block diameter,

$h_2=14$ for the M2 group classification.

$$D_{bl} \geq 14 \cdot 6,2 = 87 \text{ [mm]}$$

Due to the nominal diameters we should choose D_{bl} as 160 [mm].

The final dimensions of the traverse arm will be following: $d_0=48$ [mm]; $b=85$ mm, taking into account that it will be located under the nut of bearing №8210H due to the [18] with external diameter 80 mm. Figure 2.4 shows the main elements of the hook assembly.

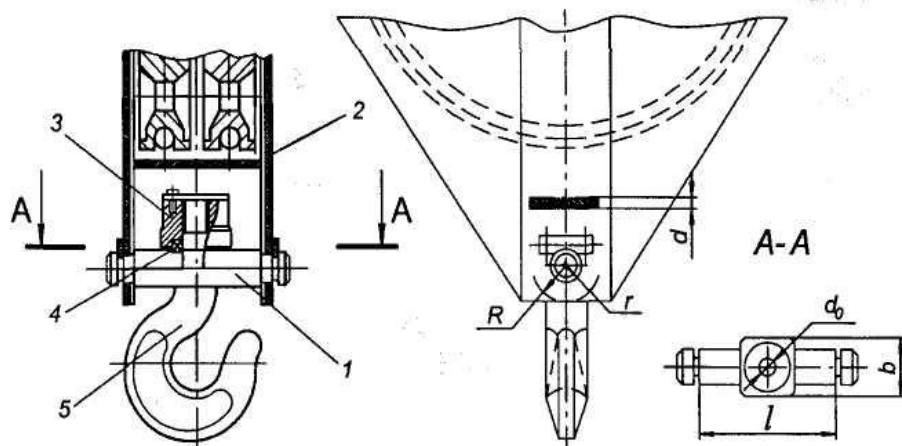


Figure 2.4 - The hook assembly view: 1 - hook traverse; 2 - hook clip; 3 – nut of the end of hook; 4 – guide bearings; 5 – hook.

For the traverse material we choose steel 45. The steel 45 is one of the most highly sought and popular types of structural carbon steel. With the help of following formula we can determine the height of the traverse:

$$h = \sqrt{\frac{3Gl}{2(b-d_0)[\sigma]}} \cdot$$

Where $[\sigma]$ allowable stress, for steel 45 it equals 120 MPa. It is calculated by the formula $[\sigma]=\sigma_T/s$, where $\sigma_T=360$ MPa - yield stress and $s=3$ yeild factor.

$$h = \sqrt{\frac{3 \cdot 50 \cdot 10^3 \cdot 100}{2 \cdot (85 - 48) \cdot 120}} = 41 \text{ [mm]}$$

Finally we rounded this value and choose $h = 45 \text{ mm}$.

The dimensions of the side was of the hook assembly will be $\delta=10 \text{ mm}$, $r=0.5$; $h=0,5 \cdot 45=22,5 \text{ mm}$; $R=90 \text{ mm}$.

It is necessary to check the strength of assembly.

$$\sigma = \frac{G}{4\delta r} \cdot \frac{R^2 + r^2}{R^2 - r^2} \leq [\sigma]$$

The allowable stress for the steel 30 XГCA $[\sigma]=\sigma_T/s=230/3,5=66 \text{ MPa}$.

Steel 30 XГCA - high-quality structural alloy steel, which is mainly used in the state after heat treatment.

$$\sigma = \frac{50 \cdot 10^3}{4 \cdot 10 \cdot 22,5} \cdot \frac{90^2 + 22,5^2}{90^2 - 22,5^2} = 62,96 \leq [\sigma] = 66 \text{ [MPa]}$$

Also, we need to check the static load capacity C_0 for the locating bearing.

$$1,25G=1,25 \cdot 50=62,5 \text{ кН} < C_0=108 \text{ [кН]}$$

The static strength was provided.

2.4 The sheave and drum design

Drum is a hollow cylindrical structure, covered on the outside with special channels – where the rope is located. The size of the groove depends on the diameter of the rope. The number of grooves depends on the number of stacked rope. If the telfers with higher lifting heights have a longer cargo rope, the drum length for such device will be longer. The diameter of the drum depends on the diameter of the cargo rope and the lift capacity of the telfer.

The drum is one of the most wear-resistant parts in the telphe. Typical drum is shown in the picture.

A balance sheave is a simple mechanism in the form of a wheel with a groove in a circle through which a flexible rope is thrown.

Blocks are divided into driving and guiding. Driving blocks are used to transmit torque from one shaft to another, guide blocks are used to change the direction of chains and ropes when lifting the loads.

The diameter of the block is calculating as follows:

$$D_{rbl} \geq h_3 d_{\kappa}.$$

Where h_3 is a diameter coefficient of the rope balancer.

The value of $h_3=12,5$ presented in the table 2.5 for the M2 mechanism classification.

Table 2.5 - Diameter coefficients for the drum

Classification of mechanism	Diameter coefficient		
	drum h_1	sheave h_2	balancer sheave h_3
M1	11.2	12.5	11.2
M2	12.5	14.0	12.5
M3	14.0	16.0	12.5
M4	16.0	18.0	14.0
M5	18.0	20.0	14.0

$$D_{rbl} \geq 12,5 \cdot 6,2 = 78 [\text{mm}]$$

Due to the nominal diameter we choose: $D_{rbl}=160\text{mm}$. The depth of groove for sheave $h=(1,4 \dots 1,9)6,2=8,7 \dots 11,8 \text{ mm}$.

The radius of groove base $R=(0,53 \dots 0,56)6,2=3,3 \dots 3,5 [\text{mm}]$

Due to this calculation we choose the depth of groove 10mm and the radius 3.5mm.

Now it is necessary to calculate the diameter of the drum. Diameter is one of the most important characteristics of the drum, and it is calculated due to the formula:

$$D \geq h_1 d_k .$$

where h_1 – coefficient of the drum diameter and it is calculated due to the table 2.5.

$h_1=12,5$ for the M2 classification of mechanisms.

$$D \geq 12,5 \cdot 6,2 = 78 \text{ [mm]}$$

As earlier for the nominal diameters of the drum we choose $D = 160\text{mm}$.

The second important characteristic of the drum is length, and it is calculated due to the formula:

$$l_d = Z_{n.r.} l_n + l_1 + 2l_0 + (Z_{n.r.} - 1)b .$$

Where l_n – length between the axes of external rope winding 1/2 of drum;

l_1 – length of drum used to attach the rope;

l_0 – length from the axes of the end wind of the rope to the edge of the drum;

b – the length of non-grooving middle part of the drum.

We need to find the distance between the axes of external rope winding:

$$l_n = t(Z_c + Z_1) .$$

Where $t \approx (1,1 \dots 1,3) d_k = (1,1 \dots 1,3) 6,2 = 6,82 \dots 8,06 \text{ mm}$, choose the middle;
 $t=7$.

Z_k – number of windings on the rope to lift the cargo for the designing height.

$Z_1 \geq 1,5$ of winding, it is the requirement which shows, that for the lower position of the cargo, the number of windings on the rope must not be less than 1.5 of winding. Due to this requirement we choose the value – 2.

$$Z_c = \frac{L_c}{\pi D}.$$

where L_c is the length of the rope, which winding on the $\frac{1}{2}$ of the drum.

$$L_c = u_b H.$$

where the H – height of lifting cargo;

u_b – mechanical advantage of block and tuckle.

So we have the following results:

$$L_c = 4 \cdot 3.5 = 14 \text{ [mm]}$$

$$Z_c = \frac{14 \cdot 10^3}{3,14 \cdot 160} = 28$$

$$l_n = 7(28 + 2) = 210 \text{ [mm]}$$

$$l_1 = 3t = 3 \cdot 7 = 21 \text{ [mm]}$$

$$l_0 = 2t = 2 \cdot 7 = 14 \text{ [mm]}$$

The next requirement should be followed:

$$b \geq B_3 - 2h_{\min} \operatorname{tg} \alpha.$$

where B_3 – the length of the external sheaves of the hook assembly,

$B_3 = 50 \text{ mm}$;

h_{\min} – minimal length between the axis of the drum and axis of the sheaves of hook assembly, $h_{\min} = 600 \text{ mm}$;

α – the maximum angle of rope inclination from normal vector of the rope to the axis of the drum, it should be $\alpha \leq 6^\circ$, we choose $\alpha = 1^\circ$.

$$b \geq 50 - 2 \cdot 600 \operatorname{tg} 1^\circ = 29 \text{ [mm]}$$

$$l_d = 2 \cdot 210 + 21 + 2 \cdot 14 + (2 - 1) \cdot 29 = 498 \text{ [mm]}$$

For the drum material we choose steel 35Л. This steel is used for the manufacture of parts operating under the influence of medium dynamic / static loads - gears, brackets, rods, balancers, latches, rollers, diaphragms, rolls, rolling mills.

2.5. Strength analysis of the drum axis

To calculate the strength characteristics of the drum it is necessary to build the scheme of the drum with bending stresses from the ropes (F_{\max}) is shown on the figure 2.5.

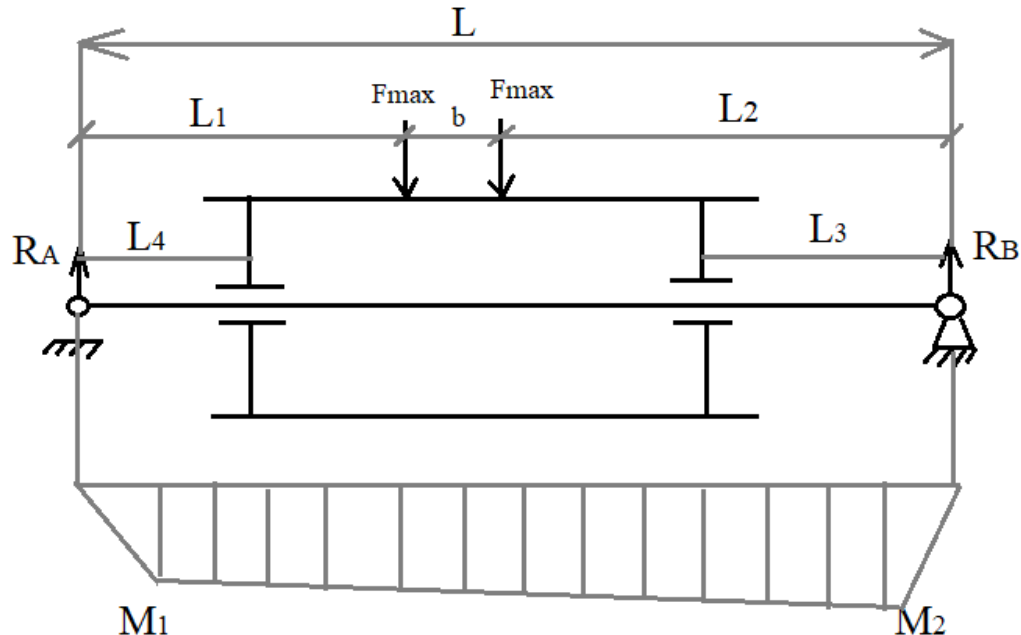


Figure 2.5 - Designing scheme of drum axis and loading

We should find following geometrical characteristics:

$$l_2 = l_n + l_0 + l_1 = 210 + 14 + 21 = 245 \text{ [mm]}$$

$$l_3=l_4=50 \text{ mm},$$

$$l = l_d + l_3 + l_4 = 498 + 50 + 50 = 598 \text{ [mm]}$$

The reactions are calculated as follows:

$$R_A = \frac{F_{\max} [l_2 + (l_2 + b)]}{l}.$$

$$R_B = 2F_{\max} - R_A.$$

$$R_A = \frac{6.9 [245 + (245 + 29)]}{598} = 5.98 \text{ [kN]}$$

$$R_B = 2 \cdot 6.9 - 5.98 = 7.82 \text{ [kN]}$$

The bending moment is calculated due to:

$$M_1 = R_A l_4$$

$$M_2 = R_B l_3$$

$$M_1 = 5.98 \cdot 0,05 = 0,29 [\text{kN} \cdot \text{m}]$$

$$M_2 = 7,82 \cdot 0,05 = 0,39 [\text{kN} \cdot \text{m}]$$

The steel 45 was choosed for the drum axis with durability limit $\sigma_{-1}=257 \text{ MPa}$.

$$[\sigma] = \frac{\sigma_{-1}}{k_0 [S]}.$$

k_0 – the coefficient for detail construction. For the axis it equals 2,0...2.8;

$[S]$ – allowable safety factor.

We choosed $k_0 - 2.5$.

$[S]$ – allowable safety factor. For the classification group of mechanisms M1...M5 $[S]=1,4$;

$$[\sigma] = \frac{257}{2,5 \cdot 1,4} = 73,4$$

The diameter of the axis in the most dangerous section of the drum under right section is calculated as follows:

$$d \geq \sqrt[3]{\frac{M}{0,1[\sigma]}}$$

$$d \geq \sqrt[3]{\frac{0,39 \cdot 10^6}{0,1 \cdot 73,4}} = 37,5 \text{ [mm]}$$

Due to this information we can assume that the diameter of the bearings due to [20].

We choose the diameter of the bearings = 40mm. It is the №1308 with static cargo capacity 8.8kN.

The radial loading for this bearing $R_B = 7,82 \text{ kN} < C_0 = 8,8 \text{ kN}$, due to this conditions the static strength was provided.

2.6 Calculation of the braking device

The braking moment should be calculated as follows:

$$T_{b.m} \geq T_{b.r} = K_b T_{st}$$

where K_{st} – coefficient of safety braking, for the M2 group classification of mechanisms $K_{st} = 1,5$;

T_{st} – static torque moment of the cargo.

$$T_{st} = \frac{GD\eta}{2u_{mech}}$$

where η – mechanism efficiency, $= 0,85$;

u_{mech} – gear reduction rate, $u_{mech} = 16$. (including the block and tuckle).

$$T_{st} = \frac{50 \cdot 0,1610,85}{2 \cdot 16} = 0,21 \text{ [kN} \cdot \text{m]}$$

$$T_{b.r.} = K_b T_{st} = 0.315 \text{ [kN} \cdot \text{m]}$$

For the calculated braking moment we need to choose the brakes, which nominal braking moment will be bigger than 0.315kN*m.

2.7. Electric engine parameters

The maximum statical power for the lifting of the cargo is calculated due the next formula:

$$P_{st \max} \approx \frac{Gv}{\eta}$$

$\eta = 0,85$ – assumed efficiency of the mechanism.

$$P_{st \max} = \frac{50 \cdot 20}{60 \cdot 0,85} = 19,6 \text{ [kW]}$$

The required power is calculated as follows:

$$P_{ep} = 0,75 P_{st \max} = P_{ep} = 0,75 P_{st \max} = 14.7 \text{ [kW]}$$

So, the it is necessary to choose the electric engine with power more or equal 15 kW. For example, we choose the electric engine with following characteristics: $P_{ep} = 15 \text{ kW}$, rotational frequency $n_{ef} = 700 \text{ min}^{-1}$, flywheel effect $D^2 = 1,71 \text{ kgm}^2$

2.8 Gear box parameters

The required gear box ratio is calculated as follows:

$$u_{gr} = \frac{n_{en}}{n_{dr}} .$$

where n_{eng} , n_{dr} – rotation frequency of the engine and drum.

The drum rotation frequency is calculated due the following formula:

$$n_d = \frac{vn_{bt}}{\pi D}$$

where n_{bt} is mechanical advantage of block and tackle; D – drum diameter.

$$n_d = \frac{20 \cdot 4}{3,14 \cdot 0,16} = 159,2 [\text{min}^{-1}]$$

$$u_{gbr} = \frac{700}{159,2} = 4,39$$

The type of the speed reducer is choosed due to [21].

The highest statical moment for the drums shaft of the lifting mechanism is calculated as follows:

$$T_{\max} = \frac{F_{\max} DZ_{n.r.}}{2\eta_d}$$

Where η_d equals 0,98 it is the drum efficiency.

$$T_{\max} = \frac{6,9 \cdot 0,16 \cdot 2}{2 \cdot 0,98} = 1,13 [\text{kN} \cdot \text{m}]$$

The equivalent moment on the output shaft of the speed reducer is calculsted as shown:

$$T_{equiv} = K_l T_{\max}$$

Where $K_l=0,5$ – lifetime coefficient.

$$T_{equiv} = 0,5 \cdot 1,13 = 0,565 [\text{kN} \cdot \text{m}]$$

So, due to the results of calculation we can choose the speed reducer from the IQY-200 family, with the reduction gear rate $n_{rg}=4$, with rotational moment $T_r=2000 \text{ N} \cdot \text{m}$.

The choosed speed reducer parameters are correct, we can choose it by the help of following relation: $T_{equiv}=565 \text{ N}\cdot\text{m} < T_r=2000 \text{ [N}\cdot\text{m]}$.

2.9 Calculation of the attachment of the rope to the drum

There are some principals how the rope can be attached to the drum. The main requirements for the type of attaching is easy in manufacture, easy procedure for rope inspection and replacement, reliability and to avoid sharp bending of the rope near the attachment fittings. The figure 2.6 shows the most typical type of the rope attachment to the drum. In this type of attachement the the longitudinal and lateral loads will not change the cross section of the rope, the grooves in the drum is non-elastic surface, the rope for this scheme is consider as smooth cylinder.

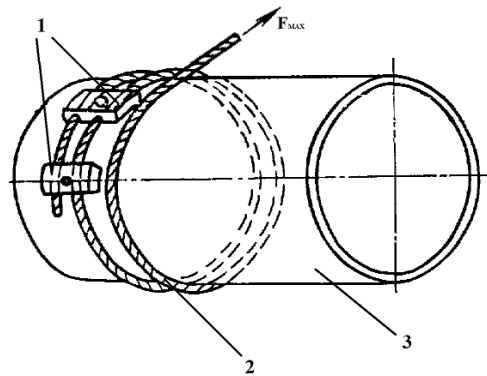


Figure 2.6 - Rope attachment: 1 - are the plates which attached the rope to the drum, 2 - is a rope, 3 - is a drum.

The rope attachment is calculated due to the formula:

$$F_0 = \frac{F_{\max}}{e^{f\alpha}}.$$

where $f=0,1$ coefficient of friction on the attached surfaces of the rope, drum and attaching plates.

α - is the wrapping angle of rope to the drum. Due to the standarts it equals 4π .

$$F_0 = \frac{6,9}{e^{0,1 \cdot 4\pi}} = 1,97 \text{ [kN]}$$

The torque bolt force is calculated as:

$$F_b = \frac{F_0}{2f} = \frac{1,97}{2 \cdot 0,1} = 9,85$$

The diameters of the bolt which attached the plates approximately equal 7mm (d_b) due to the M2 classification of the mechanisms.

And the quantity = 4 bolts(z_b).

The stress which appears in the bolts in the attachment point is calculates as:

$$\sigma = \frac{1,3F_b}{0,25z_b\pi d_b^2} + \frac{1,5F_0d}{z_b 0,1d_b^3} \leq [\sigma]$$

$$\sigma = \frac{1,3 \cdot 9,85 \cdot 10^3}{0,25 \cdot 4 \cdot 3,14 \cdot 8^2} + \frac{1,5 \cdot 1,97 \cdot 10^3 \cdot 8}{4 \cdot 0,1 \cdot 8^3} = 179.$$

The allowable stresses for the bolts which are made from the steel 35 is $[\sigma] = 0,6\sigma_m = 0,6 \cdot 320 = 192 \text{ MPa}$.

Steel 35 - structural carbon, used in the manufacture of parts of low strength.

The required condition is satisfied.

2.10 Movable parts of the lifting mechanism

The movement of the trolley is provided by the four wheels. The two of them are located in the forward part and the next two are located in the rear part. These wheels have a complex structure. There is a working part of the wheels, which contacts with the rail and this surface is take all loads, and there is a special borders in each wheel. The function of this border is to direct along the rail.

The weight of movable mechanism for all M2 class of mechanisms are calculated as follows:

$$G_{movablemech} = (0,25 \dots 0,35)G$$

$$G_{movablemech} = (0,25 \dots 0,35) \cdot 50 = 12,5 \dots 17,5 [kN]$$

Chooosed weight of movable device is 15 kN. The diameter of the whels is choosen due to the formula:

$$F_{st.max} \leq [F_{w.max}] .$$

where $F_{st.max}$ is a maximum statical load on the wheel.

$[F_{w.max}]$ is allowable load on the one wheel.

$$F_{st.max} = \frac{G_{movablemech} + G}{n} .$$

The n is a number of wheels in the mechanism = 4.

$$F_{st.max} = \frac{15 + 50}{4} = 16,25 \text{ [kN]}$$

For the $[F_{w.max}] = 25\text{kN}$, we choose the diameter of the wheels

$$D_w = 200\text{mm}.$$

The rotational frequency of the wheels, can be calculated as follows:

$$n_w = \frac{v}{\pi \cdot D_w} .$$

where v is a movable speed = 25.

$$n_w = \frac{25}{3,14 \cdot 0,2} = 39,8 \text{ [min}^{-1}\text{]}$$

Due to our calculations the actual speed of the movable mechanism will be:

$$v_a = \pi n_w D_w .$$

$$v_a = 3,14 \cdot 39,8 \cdot 0,2 = 25[\text{m} / \text{min}]$$

So, the all necessary parameters of the telpher were calculated. The designed requirements are satisfied.

Conclusion to the part

In this part of diploma work we have calculated the main design parameters of the upper loading device – telfer. We designed the telfer with cargo capacity up to 5 tons. This telfer will be attached for the special mono rail. The maximum height for cargo lifting is 3.5 meters. This height satisfies our needs, because the diameter of the fuselage is 3.35 meters. The lifting speed of our cargo device will be 20 m/min with horizontal speed movement up to 25m/min.

The telfer will move from the front part of the cargo cabin to the rear part.

It can attach the cargo with the help of designed hook. The telfer will lift all types of cargo, which will have the special fasteners. The hook can attach the cargo which is located on the cargo ramp. There is a special mechanical loader needed to move the cargo from the ground to the aircraft ramp.

GENERAL CONCLUSIONS

In the scope of diploma work we have the following results:

- preliminary design of short range cargo aircraft with cargo capacity 16 tons was performed

- the centre of gravity positions were defined;
- the engines with high efficiency were choosed;
- the upper loading device telpher was designed;
- all necessary parametrs were choosed for the telpher.

The scheme designed high wing aircraft with two engines under the wing was choosed. The Д436Т1 engine was choosed for our aircraft. This engine meets all requirements of the International Civil Aviation Organization (ICAO). There is a less amount of emmsions on the atmosphere and less noise pollution gives possibility to use this aircraft in European countries. Also, high wing schem improves protection for the engines from the dirt and harmful things. So, due to this the aircraft can be used in non-prepared runways.

The loading and unloding of the cargo will be provided by the help of telpher.

This upper loading device have the cargo capacity up to 5 tons and with maxim heigt of lift 3.5 meters.

The telpher is mounted on the mono-rail which is located under the ceiling. The telpher gives an opportunity to move the different types cargo in vertical and horizontal directions.

The designed aircraft combines most optimal aerodynamic characteristics, design factors, modern materials which improves aircraft's efficiency. Properly selected these parameters will deacreases the fuel consumption which nowadays is the most important factor in the aircraft industry.

<i>Department of Aircraft Design</i>				<i>NAU 20 09 L 00 00 00 59 EN</i>			
<i>Performed by</i>	<i>Lutovinov M.R</i>			<i>General conclusions</i>	<i>Letter</i>	<i>Sheet</i>	<i>Sheets</i>
<i>Principal</i>	<i>Maslak T.P.</i>						
<i>Adviser</i>					<i>402 AF 134</i>		
<i>Stand.contr.</i>	<i>Khizhnyak S.V</i>						
<i>Head of dep.</i>	<i>Ignatovych S.R.</i>						

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<i>Department of Aircraft Design</i>				<i>NAU 20 09 L 00 00 00 59 EN</i>			
<i>Performed by</i>	<i>Lutovinov M.R</i>			<i>References</i>	<i>Letter</i>	<i>Sheet</i>	<i>Sheets</i>
<i>Principal</i>	<i>Maslak T,P</i>						
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<i>Head of dep.</i>	<i>Ignatovych S.R.</i>						
					<i>402 AF 134</i>		

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Appendix A

ПРОЕКТ
САМОЛЕТА СТРДД
НАУ, АКИ, кафедра КЛА

ПРОЕКТ DIPLOMA Расчет выполнен 23.09.2019
Исполнитель LUTOVINOV Руководитель MASLAK T.P

ИСХОДНЫЕ ДАННЫЕ И ВЫБРАННЫЕ ПАРАМЕТРЫ

Количество пассажиров	0.
Количество членов экипажа	2.
Количество бортпроводников или сопровождающих	2.
Масса снаряжения и служебного груза	896.66 кг.
Масса коммерческой нагрузки	16000.00 кг.
Крейсерская скорость полета	800. км/ч
Число "М" полета при крейсерской скорости	0.7491
Расчетная высота начала реализации полетов с крейсерской экономической скоростью	11.000 км
Дальность полета с максимальной коммерческой нагрузкой	1300. км.
Длина летной полосы аэродрома базирования	2.55 км.
Количество двигателей	2.
Оценка по статистике тяговооруженности	3.2000 Н/кг
Степень повышения давления	5.50
Относительная масса топлива по статистике	0.3100
Удлинение крыла	9.58
Сужение крыла	4.05
Средняя относительная толщина крыла	0.118
Стреловидность крыла по 0.25 хорд	24.0 град.
Степень механизированности крыла	0.970
Относительная площадь прикорневых наплывов	0.000
Профиль крыла – Суперкритический	
Шайбы УИТКОМБА – не применяются	
Спойлеры – установлены	
Диаметр фюзеляжа	3.35 м.
Удлинение фюзеляжа	8.00
Стреловидность горизонтального оперения	32.0 град.
Стреловидность вертикального оперения	40.0 град.

РЕЗУЛЬТАТЫ РАСЧЕТА
НА У, АК И, КАФЕДРА "К Л А"

Значение оптимального коэффициента под'емной силы в расчетной точке крейсерского режима полета	C_y	0.43976
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Значение коэффициента	Сх.инд.	0.00916
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ОПРЕДЕЛЕНИЕ КОЭФФИЦИЕНТА $D_m = M_{\text{крит}} - M_{\text{крейс}}$

Число Маха крейсерское	Мкрейс	0.74974
Число Маха волнового кризиса	Мкрит	0.76808
Вычисленное значение	Dm	0.01833

Значения удельных нагрузок на крыло в кПА (по полной площади) :

при взлете	4.223
в середине крейсерского участка	3.937
в начале крейсерского участка	4.068

Значение коэффициента сопротивления фюзеляжа и гондол	0.00847
Значение коэфф. профиль. сопротивления крыла и оперения	0.000917
Значение коэффициента сопротивления самолета:	
в начале крейсерского режима	0.02820
в середине крейсерского режима	0.02789

Среднее значение C_y при условном полете по потолкам	0.43976
Среднее крейсерское качество самолета	15.76555

Значение коэффициента $C_{y.пос.}$	1.631
Значение коэффициента (при скорости сваливания) $C_{y.пос.макс.}$	2.447
Значение коэффициента (при скорости сваливания) $C_{y.взл.макс.}$	2.039
Значение коэффициента $C_{y.отр.}$	1.488
Энерговооруженность в начале крейсерского режима	0.621
Стартовая тяговооруженность. по условиям крейс. режима $R_{o.кр.}$	2.608
Стартовая тяговоруж. по условиям безопасного взлета $R_{o.взл.}$	2.487

Расчетная тяговооруженность самолета N_o 2.713

Отношение $D_n = R_{o.кр.} / R_{o.взл.}$ D_r 1.049

УДЕЛЬНЫЕ РАСХОДЫ ТОПЛИВА (в кг/кВт*ч) :

взлетный	37.1755
крейсерский (характеристика двигателя)	58.1239
средний крейсерский при заданной дальности полета	58.6662

ОТНОСИТЕЛЬНЫЕ МАССЫ ТОПЛИВА:

аэронавигационный запас	0.03349
расходуемая масса топлива	0.09739

ЗНАЧЕНИЯ ОТНОСИТЕЛЬНЫХ МАСС:

крыла	0.11993
горизонтального оперения	0.01747
вертикального оперения	0.02031
шасси	0.04959
силовой установки	0.08915
фюзеляжа	0.12476
оборудования и управления	0.12984
служебной нагрузки	0.01688
топлива при $L_{расч.}$	0.13088
коммерческой нагрузки	0.30120

Взлетная масса самолета " M_o " = 53122. кг.
Потребная взлетная тяга двигателя 72.05 кН

Относительная масса высотного оборудования и противообледенительной системы самолета	0.0171
Относительная масса пассажирского оборудования (или оборудования кабин грузового самолета)	0.0006
Относительная масса декоративной обшивки и ТЗИ	0.0062
Относительная масса бытового (или грузового) оборудования	0.02845
Относительная масса управления	0.0073
Относительная масса гидросистем	0.0194

Относительная масса электрооборудования	0.0208
Относительная масса локационного оборудования	0.0059
Относительная масса навигационного оборудования	0.0088
Относительная масса радиосвязного оборудования	0.0044
Относительная масса приборного оборудования	0.0103
Относительная масса топливной системы (входит в массу "СУ")	0.0038

Дополнительное оснащение:

Относительная масса контейнерного оборудования	0.0000
Относительная масса нетипичного оборудования	0.0037
[встроенные системы диагностики и контроля параметров, дополнительное оснащение салонов и пр.]	

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ

Скорость отрыва самолета	242.45 км/ч
Ускорение при разбеге	2.07 м/с*с
Длина разбега самолета	1094. м.
Дистанция набора безопасной высоты	578. м.
Взлетная дистанция	1678. м.

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ

ПРОДОЛЖЕННОГО ВЗЛЕТА

Скорость принятия решения	230.32 км/ч
Среднее ускорение при продолженном взлете на мокрой ВПП	0.22 м/с*с
Длина разбега при продолженном взлете на мокрой ВПП	1973.98 м.
Взлетная дистанция продолженного взлета	2552.36 м.
Потребная длина летной полосы по условиям прерванного взлета	2646.29 м.

ХАРАКТЕРИСТИКИ ПОСАДОЧНОЙ ДИСТАНЦИИ

Максимальная посадочная масса самолета	50399. кг.
Время снижения с высоты эшелона до высоты полета по кругу	21.5 мин.
Дистанция снижения	47.74 км.
Скорость захода на посадку	242.55 км/ч.
Средняя вертикальная скорость снижения	1.97 м/с
Дистанция воздушного участка	514. м.
Посадочная скорость	227.55 км/ч.
Длина пробега	714. м.
Посадочная дистанция	1228. м.
Потребная длина летной полосы (ВПП + КПВ) для основного аэродрома	2051. м.
Потребная длина летной полосы для запасного аэродрома	1744. м.

ПОКАЗАТЕЛИ ЭФФЕКТИВНОСТИ САМОЛЕТА

Отношение массы снаряженного самолета к массе коммерческой нагрузки	1.8737
Масса пустого снаряженного с-та приход. на 1 пассажира	0.00 кг/пас.
Относительная производительность по полной нагрузке	345.66 км/ч
Производительность с-та при макс. коммерч. нагрузке	10805.3 кг*км/ч
Средний часовой расход топлива	2687.411 кг/ч
Средний километровый расход топлива	3.98 кг/км
Средний расход топлива на тоннокилометр	248.715 г/(т*км)
Средний расход топлива на пассажирокилометр	0.0000 г/(пас.*км)
Ориентировочная оценка приведен. затрат на тоннокилометр	0.2038 \$/(т*км)

Appendix B

